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RA-1 INTERIM REPORT

SPACECRAFT FLIGHT PERFORMANCE

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RA-1 SPACECRAFT FLIGHT PERFORMANCE

INTRODUCTION by A. E. Dickinson - Division 31

This report summarizes the in-flight performance of the Ranger A-1 Spacecraft. Section I is devoted to a flight summary which describes certain aspects of the trajectory, the flight sequence, and spacecraft telemetry which are of particular importance in determining how the various systems of the spacecraft actually performed during the lifetime of the mission. Section II summarizes the performance of each of the major spacecraft systems and certain subsystems as well as the engineering experiments. The following are not within the scope of this report: launch vehicle performance, tracking and orbit determination data, and scientific experiment data.

I. FLIGHT SUMMARY by A. E. Dickinson - Division 31

Ranger A-1 was launched from AMR Cape Canaveral on August 23, 1961 at 10.26 seconds after 5:04 A.M., E.S.T. The GMT of liftoff was 235:10:04:10.26 Z (All subsequent references to time will be in GMT.) The ascent phase and injection into parking orbit were normal. The spacecraft shroud was ejected as programmed. The spacecraft transponder was kept in two-way lock with the Launch Checkout Telemetry Trailer (LCTT) for 7½ minutes after liftoff except for an unexplained momentary dropout.

The Agena restart sequence began at the proper time but the fuel tank valve failed to operate and ignition did not occur. The result of this was the expulsion of sufficient oxidizer gas to add a velocity increment of approximately 70 meters per second. The subsequent electrical and mechanical separation of spacecraft from the Agena appears to have been normal. The spacecraft was injected at 102723Z into a satellite orbit with a period of 91.15 minutes and apogee and perigee of 313.3 and 105.6 statute miles respectively. By August 28 (4.6 days later), the last day on which the spacecraft signal was tracked, the orbit had decayed to a period of 89.19 minutes with apogee and perigee of 202.9 and 96.2 statute miles respectively.

The spacecraft controller is programmed to issue a sequence of ten discrete commands to different units in the spacecraft within six hours after injection. A list of these commands and the programmed times is given in the following table:

Number	Command	Time from Liftoff
1	Increase transmitter power	30.3 min.
2	Turn on scientific instruments high voltage	78.7 "
3	Extend particle analyzer boom and solar panels	33.6 "
4	Start sun acquisition	58.7 "
5	Start earth acquisition	90.3 "
6	Change data encoder rate gyro scale	115.3 "
7	Open semiconductor detector aperture	203.7 "
8	Switch transponder transmitter to hi-gain antenna	247.0 "
9	Reduce modulation on beacon transmitter	363.7 "
10	Turn on friction-lubrication experiment	367.0 "

The spacecraft can also accept the following commands from the ground as a backup for controller commands 5 and 8 or to correct certain non-standard conditions:

1. Roll override
2. Hinge override
3. Antenna transfer (cyclical)

Controller command 2 was programmed at the time indicated as the result of an incident which occurred during the countdown on August 3. When this command was sent from the blockhouse, it apparently caused the controller to issue all the other programmed commands. Subsequently, this step was eliminated from the countdown procedure and the time of the programmed command delayed in order to minimize the effect in case of a similar occurrence during flight.

Telemetry records indicate that all programmed events took place. The nature of the orbit was such that it is impossible to determine the exact time of any of the programmed commands. Command 1 occurred before the Mobile Tracking Station (DSIF 1) in South Africa acquired the spacecraft signal. At the time of command 3 the signal was too noisy. All the other commands occurred during periods when the spacecraft was not in view of any tracking station. In each case, however, subsequent telemetry indicated that each commanded function did occur within the proper time interval.

Probably the most serious effect of the satellite trajectory was the fact that the spacecraft was in the earth's shadow about 45% of the time during each orbital revolution. Each time the spacecraft returned to the sunlight, the attitude control system, if operating normally, would commence its acquisition procedure with a consequent heavy drain on the supply of nitrogen gas used for activation. This gas supply was exhausted within 19 hours after injection, sometime after the 5th orbit (the last orbit tracked on August 23) and before the 13th orbit (the first orbit tracked on August 24.)

A number of apparently related conditions appeared with the first data obtained on August 24, lasted throughout all passes tracked on that day (up to the 21st orbit) and did not show up again after that. (1) The attitude control (A/C) power converter was overloaded, indicated by the converter monitor voltage reading of zero. (2) Spikes appeared on the analog trace of all the attitude control measurements. (3) Except for the spikes, all the attitude control measurements (rates, positions, and light detector) had null values (due to the absence of dc power). (4) The 400 cps tone on Channel 1 exhibited a considerable degree of what can best be described as amplitude modulation. The "modulation" was quite variable, especially in amplitude, with at least two identifiable frequency components present at times. (5) The data encoder commutator was stepped faster than normal by the controller generally increasing its speed to a maximum error of about 4% on the 15th and 16th orbits, then decreasing back to normal by the 20th orbit. This apparently was done without skipping any segments. The Lyman Alpha telescope scanning times were also affected and it is quite likely the scientific data automation system (DAS) was similarly affected but this has not yet been extablished.

The frequency of the spikes, occurring every .41 to .42 seconds, corresponds to drops in amplitude of the 400 cps signal. It is believed that both phenomena resulted from the A/C converter overload interrogation process. The cause of some of the other things which were happening is still not known. It would seem logical to assume that the excessive current requirements in the A/C system which caused the converter to go into overload are the proximate result of loss of nitrogen gas but this cannot be proved. Nor does it explain why the overload condition subsequently disappeared.

The ground-to-spacecraft command link was successfully tested on August 25. The ground transmitter at Goldstone Az-El site (DSIF 3) switched the transponder transmitter from the hi-gain to the low-gain omnidirectional antenna during the 30th orbit. During the following orbit the same station switched the 3 watt transmitter to the hi-gain antenna and back again. Then the LCTT transmitter switched it back to the hi-gain antenna where it remained for the rest of the flight. The reason for leaving it on the hi-gain antenna was that the received signal level at the DSIF stations was usually higher in this mode, even when locked on a side lobe, than it was when the transponder was transmitting over the lo-gain antenna with its minus 10 db coupler.

On August 26 the Hinge and Roll Override commands were transmitted by DSIF 2 at Goldstone but there was no way to determine their effect on the spacecraft since the spacecraft was not stabilized and the rate gyros were usually saturated.

The first indication of the main battery failing was on the 61st orbit on August 27, approximately 91 hours after liftoff. The last good telemetry was recorded 40 minutes later at 0549Z. After that the quarter-watt transmitter or beacon, which has its own battery power supply, continued to operate, usually with no modulation, until after the 83rd orbit, the last pass tracked on August 28. During this period sporadic modulation appeared on the beacon signal when the solar panels were supplying enough current to activate the data encoding system. The transponder signal was occasionally picked up at these times. The last signal received from the spacecraft was a momentary one from the transponder at 235336Z on August 28 during the 89th orbit. Apparently the beacon battery was dead by then.

The orbit of Ranger 1 created serious problems in recovering and reducing telemetry data. Of a total of 36 spacecraft engineering measurements, all of which are sampled by commutation, 23 are sampled by the data encoder only once every 1000 seconds. All of the power system and temperature measurements are included in this group. Proper reduction of these telemetered measurements for evaluation requires that the ground decommutator be synchronized with the spacecraft commutators. This is done by using the 400 cps tone on

1R1G Channel 1 and synchronization tones for each of the three commutator speeds. Since the sync tone for the slowest rate measurements (Rate 3) is sampled only once every 1000 seconds, and because the station passes were always less than 10 minutes or 600 seconds, the ground decommutators were seldom synchronized with the spacecraft rate 3 commutator decks. As a result these samples usually had to be decommutated by hand.

During the 92 hour lifetime of the main battery (through 61 orbital revolutions) portions of 39 orbits were tracked by the DSIF stations and the LCTT. The total tracking coverage of approximately 500 minutes includes all the periods when the spacecraft signal was lost or below receiver threshold in the middle of a pass. A reasonable estimate of the total amount of useful telemetry obtained would be 300-400 minutes. The following table shows a breakdown of this coverage by days:

Date	Orbits	Total time of tracking coverage (maximum)	Stations
Aug. 23	1-5	41 minutes	DSIF 1,2,3
24	13-21	109 "	DSIF 1,2,3,(5),LCTT
25	27-36	145 "	DSIF 1,2,3,4,(5),LCTT
26	42-52	151 "	DSIF 1,2,3,4,5, LCTT
27	58-61	53 " (before battery failure)	DSIF 2,3,4, LCTT
		Total 499 minutes (maximum)	

II. SUBSYSTEM PERFORMANCE SUMMARY by A. E. Dickinson - Division 31

1. General System Performance. The overall performance of the Ranger A1 spacecraft was successful considering the nature of the orbit flown. In most cases where an element of the spacecraft did not perform as it was designed to do, it was because the particular element was not designed to operate in a near-earth environment. No malfunctions were detected that would warrant making any design or hardware changes in Ranger A-2.

2. Space Sciences Subsystems. The value of most of the scientific experiments was seriously reduced because of proximity to the earth and lack of attitude stabilization. The DAS exhibited occasional minor malfunctions such as premature program reset, data register reset and extra counts in the frame-count register. The scientific instruments in general performed as expected considering their operating environment. The magnetometer was saturated by the earth's field throughout the mission.

3. Telecommunications Subsystems. All available information indicates that all elements of spacecraft communications system performed within their design tolerances. This includes the transponder, data encoder, command system, and quarter-watt transmitter. The high gain antenna drive indicated a higher power output than specified when the transponder was switched on to the high gain antenna.

4. Guidance and Control Subsystems.

A. Controller. The spacecraft controller apparently issued all programmed commands in the proper sequence but the exact times cannot be verified. It also provided a 1 pulse per second signal to the data encoder for commutator stepping with a high degree of accuracy except on the second day of flight when the stepping rate increased. The same is true of the 1 pulse per 1000 seconds provided to the Lyman Alpha telescope. The accuracy of the 1 pulse per second signal to other scientific instruments has not been throughly checked.

B. Attitude Control System. The performance of this system is difficult to evaluate because of the limited amount of telemetry obtained before gas depletion. Sun acquisition, as indicated by solar panel current measurements, was only observed on one pass. On other occasions it appeared that the earth sensor had locked on the sun. No valid conclusions can be drawn concerning limit cycle performance, antenna pointing accuracy or gas consumption rate. Measurements of the rate gyros showed that they were usually saturated due to spacecraft tumbling after the nitrogen gas was gone.

C. Central Power System. Because the spacecraft was seldom oriented toward the sun, the solar panels provided only intermittent power, resulting in an almost continuous drain on the launch back-up battery. This main battery's lifetime of about 92 hours is considered normal in view of the system power demands. The few correlated measurements of panel current and temperature available indicate normal performance under the circumstances prevailing.

The power conversion and distribution system apparently functioned normally. The overload in the attitude control dc to dc converter on Aug. 24 was probably caused by excessive power demands in that system resulting from gas depletion and marginal overload design conditons. Some of the other symptoms previously described, which occurred simultaneously may have been caused by noise generated by the A/C converter's interrogation of its load.

D. Beacon Battery. The operating lifetime of the quarter-watt transmitter battery was less than 5.6 days, compared to a design value of 10 days. Whether this was due to a discrete failure, or the battery life was reduced by its unexpected environment, is not known.

E. Solar Cell Experiment. Not enough data was obtained to make any definite evaluation of the advanced development solar cell experiment.

5. Engineering Mechanics Subsystems.

A. Agena Interface. Shock, vibration and acceleration levels during boost phase of the mission were generally as expected. Spacecraft separation, both electrical and mechanical, was apparently normal.

B. Erections. The erection of the SCREPA boom and solar panels was apparently normal.

C. Temperature Control. Temperature measurements of elements within the main hexagonal frame of the spacecraft showed good agreement with values calculated for the actual orbit. The temperature of such "external" elements as the solar panels, earth sensor and magnetometer showed considerably more variation than the hex temperatures but this is to be expected considering the orbit.

D. Friction-Lubrication Experient. This experiment was designed to measure coefficients of friction in the near-total vacuum of space. Although the value of the data obtained was greatly reduced due to the low altitude orbit, the experiment apparently performed normally under the circumstances. Its self-contained battery power supply operated longer than it was designed to do.

III. SUBSYSTEM ANALYSIS

SCIENTIFIC INSTRUMENT AND DAS PERFORMANCE by M. Neugebauer - Division 32

1. Operation of Instruments and DAS compared to normal, taking account of the orbit

   A. The DAS operated correctly with the following occasional minor malfunctions:

      1. Complete reset of all data registers observed at the following times:

         (1) Liftoff + 124 seconds

         (2) Between end of launch station track and MTS acquisition at 1211 Z on August 23.

         (3) Between above pass and following Goldstone acquisition at 1308 Z.

         (4) Between end of 3rd MTS pass at 1355 Z on 8/23 and Goldstone pass on 8/24 at 0833 Z.

         (5) Between last MTS pass on 8/24 at 1712 Z and first LCTT pass on 8/25 at0230 Z.

      2. Program reset within the DAS (i.e., it would start a new frame without finishing the one it was on.)

      3. Addition of 32 counts to the frame-count register between passes.

      4. Malfunction of the ion-chamber time register, probably caused by noise.

   B. The ion chamber, triple coincidence telescopes, gold-silicon detector, and Geiger tube all appeared to have operated approximately as expected in the satellite environment.

   C. The cadmium-sulfide detectors gave spurious results in that supposedly matched detectors had very different counting rates. One possible explanation is that the CdS crystals each had very different time constants for recovery after looking directly at the Sun.

D. The micrometeorite detector exhibited an anomolously high light-flash counting rate when the spacecraft was in the sunlight. This was presumably due to direct or reflected sunlight and/or Earthlight. The instrument operated correctly when in the dark.

E. The electrostatic analyzers appear to have operated normally except that it is believed that there was no voltage maintained across the deflection plates. This could have been caused by the excessive collection of ionospheric electrons or ions by the outside of the deflection plates which would cause the loading down of the deflection plate power supply.

F. The Lyman-alpha telescope appeared to function normally. Part or all of five pictures and several in-flight-calibrations were observed as well as background measurements of Lyman-alpha intensity.

G. No magnetic field data were obtained both because the Earth's field at the Ranger-I altitude was higher than could be measured with this experiment and because the magnetometer temperature was usually outside the operational range.

H. The Vela Hotel experiment appeared to have operated exactly as expected considering the orbit.

2. <u>Indications of spacecraft performance as determined from the scientific data.</u>

A. There were probably certain power transients, as discussed in Part I-A above.

B. Some idea of the spacecraft orientation and/or tumble rate can be obtained by examining the Lyman-alpha and electrostatic analyzer data for passes in the dark, and also by observing the output of the light sensitive CdS and micrometeorite detectors for passes when the spacecraft was on the sunlit side of the earth. To date this analysis has been performed for only a very few passes.

C. Temperature data were obtained and are available for the ion Chamber and for the gold-silicon, cadmium-sulfide, and triple-coincidence detectors.

3. Effect of the scientific instruments of DAS on the rest of the spacecraft.

No such effects are apparent at this time.

4. Quality of data transmission from spacecraft.

There were great extremes in the quality of the scientific data, varying from completely useless noise with no recognizable frame pattern to a pass with four (4) complete data frames (8 minutes) of perfect binary data.

COMMUNICATIONS SYSTEM PERFORMANCE by R. P. Mathison - Division 33

All available information points to the fact that the spacecraft communications system functioned normally and that no degradations were apparent which would indicate that the system would not have performed its function had the trajectory been normal. The following summary indicates those parameters analyzed:

Parameter	Remarks
Transponder Temperature	Within the range of spacecraft specifications
Low gain antenna drive	all readings within tolerance, indicates antenna switchover was normal
Transponder + 250 & 150 plate volts	Indicates power up was normal
Transponder local oscillator drive	Within tolerance, indicates slight drift correlated with temperature as would be expected.
High gain antenna drive	Higher power output than specifications, indicates antenna switchover was normal
Transponder power out	Normal, indicates power up and antenna switchover was normal.

RA-1 VIBRATION AND SHOCK ENVIRONMENT by A. P. Bowman - Division 31 and J. I. McPherson - Division 35

The shock and vibration environment of the RA-1 S/C, as measured at the separation plane, is discussed in detail. The period for which data was available extends from pre-lift-off ignition up to but excluding S/C separation from the Agena.

The vibration and shock instrumentation flown on RA-1 consisted of five pickups, three Statham low frequency and two Endevco high frequency accelerometers. The Stathams had a bandwidth from DC to 100 or 150 cps depending on telemetry channel involved. The Endevco accelerometers had a bandwidth of 20 - 2000 cps. The accelerometer pickups were located at LMSD Station 232.50 which is essentially the foot of the S/C. The three Statham accelerometers were oriented in the radial (Ch 10 and 11) and lateral (Ch 12) directions. The two Endevco accelerometers were oriented in the radial (Ch 18) and axial (Ch 17) directions. The pickup location and orientation are shown in the Instrumentation Installation sketch included.

The flight data may logically be classed into two groups: steady state and transient. The steady state phenomenon has been defined as vibration; whereas, the transient phenomenon has been defined as shock. The separation of transient and steady state is based on judgement rather than on the analysis of the system based on speed of response and damping; therefore, some of the transient motion may be of a short term, steady state nature in some systems. The same statement may also be true for some steady state conditions; however, the time sample used for this steady state analysis is fairly large and the frequencies fairly high, thus making the steady state phenomenon more certain.

Steady State Vibration

The RMS g levels of Ch 17 and 18 are plotted as a function of time, in Figure 1 and 2, up to the mid-portion of 1st Agena burn. The data point marked "shroud separation" is not valid due to the type of data reduction used and should be disregarded; this data point will be discussed later under transient excitation. The maximum steady state vibration levels occurred during transonic flight shortly after lift-off (at about 19,000 ft.). The flight data from this portion of flight has been replotted from the original power spectral density plots obtained from LMSD and is represented in Figures 3 and 4 for Ch 19 and 18 respectively. The actual tape loops were taken just before the sharp g pike. In addition to the flight data on Figures 3 and 4, the predicted acoustic-free response of the S/C to the measured input and the appropriate noise portion of pertinent JPL specifications are plotted; i.e., the maximum square of the envelope of spacecraft response as measured in the environmental lab was multiplied point by point by the PSD of in-flight vibration data, thus obtaining the PSD of acoustic-free response.

The curves in Figures 3 and 4, marked "flight data," should only be compared with JPL Spec. 30216 and 30222 since these specifications are for the complete S/C and represent inputs to the feet. The curves in Figures 3 and 4 marked "composite response" should be compared with the FA and TA of Spec. 30201. It should be noted that the flight accelerometers were not located in the pitch and yaw direction but read at some angle in between. We have assumed, however, that the environment was roughly the same in pitch and yaw as in the measured axis.

The low frequency flight data from RA-1 is plotted in Figure 5. The "in-flight vibration" levels were taken from oscilograph records of Ch 10 (radial), Ch 11 (radial), and Ch 12 (lateral). The levels indicated are the maximum levels observed during flight, neglecting major transients in the data. The data represented in Figure 5 is thus the steady state excitation plus minor transient events. Plotted with this curve is the sinusoidal portion of the FA vibration test level (JPL Spec. 30222) used on RA-1 flight S/C. The curves in Figure 6 represent the maximum acoustic-free response of the S/C to the in-flight input given in Figure 5 and the low frequency sinusoidal portion of the assembly test specification 30201.

Transient Excitation

The shock spectrum of transient events which occurred during the flight of RA-1 S/C are compared with the theoretical shock spectrum of the environmental shock test per JPL Spec. 30201. The shock spectrum for the flight was obtained by playing Ch 17 and 18 magnetic records through a shock spectrum analyzer and thus obtaining the response or shock spectrum. The curve labeled "maximum envelope" was obtained by considering the individual shock spectrum of Ch 17 and 18 at each transient event and selecting the maximum readings from the numerous shock spectra. The composite curve is mainly based on two transient events, transonic and shroud separation, the other events being relatively unimportant in comparison; (this may be substantiated by observing the peaks in Figures 1 and 2). The data presented in Figure 7 is of a somewhat questionable value since Ch 17 and 18 overloaded during the shroud separation. The levels do, however, indicate a lower limit to the probable shock environment at the S/C feet. It should be noted that S/C separation, which is the most severe shock input from the Agena, has not been included. The data from this event was not reduced due to the extended overload in Ch 17 and 18. In addition since the shock spectrum in Figure 7 is the shock input to the feet of the S/C, the levels would most certainly be modified and in most cases attenuated as they proceed into the S/C structure and thus to assemblies.

A rather severe low frequency transient was noted on Ch 12 at booster cut-off. The peak reading was 2.5 g peak with a damped ring out at about 65 cps. The remaining channels (10, 11, 17, and 18) recorded at 65 cps ring out at levels ranging from 0.2 to 0.5 g. In addition, Ch 10, 12, 17, and 18 are all at the same location. The only conclusion which can be reached at this time is that Ch 12 reading of 2.5 g is not a true indication of S/c environment.

Conclusions

The general conclusion which may be drawn from the various curves is that the S/C was adequately tested for the vibration and shock levels encountered in the flight of Vehicle 6001 or RA-1. There are qualifications which must be added to clarify several points.

The comparison of low frequency flight environment for pitch and yaw with the FA test of RA-1 S/C, as depicted in Figure 5, indicates that the test was adequate. Figure 6 indicates that two or three portions of the S/C that resonate near 15 and 30 cps were inadequately tested at the assembly level. The actual levels as seen by most portions of the omni-antenna, Lyman & Screpa using the assembly spec. would be effected by a resonance gain of possibly 5 or 10. The testing of these assemblies was done in such a manner, excitation from point of attachment to spacecraft, that the vibration levels on the body of the instruments was in excess of the levels predicted here. The levels experienced by these assemblies during the S/C FA test did exceed assembly test levels and the flight environment, and, as a result, all assumblies were qualified at the assembly level and/or the S/C test level. In addition, the PTM S/C was tested to levels in excess of levels used for FA testing.

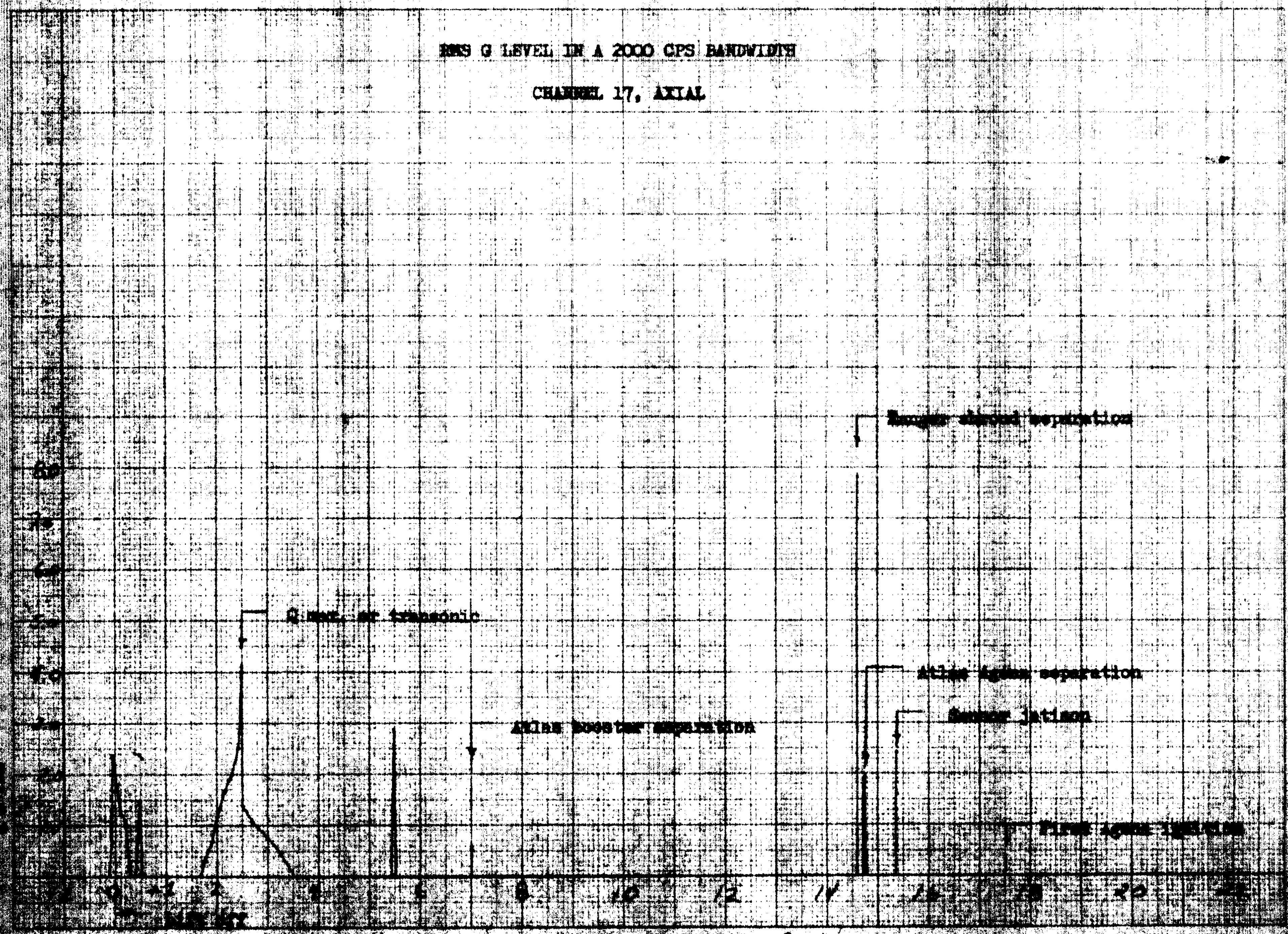
The high frequency, 150 to 1500 cps, data (Figures 3 and 4) indicates that the S/C FA levels were not exceeded in the axial direction. The in-flight vibration levels for the radial direction (pitch and yaw) exceeded

S/C FA and PTM levels; however, the excess of flight levels over test levels were not excessive (about 35% over). The vibration levels to be used for RA-3 PTM are higher and thus eliminate this difficulty.

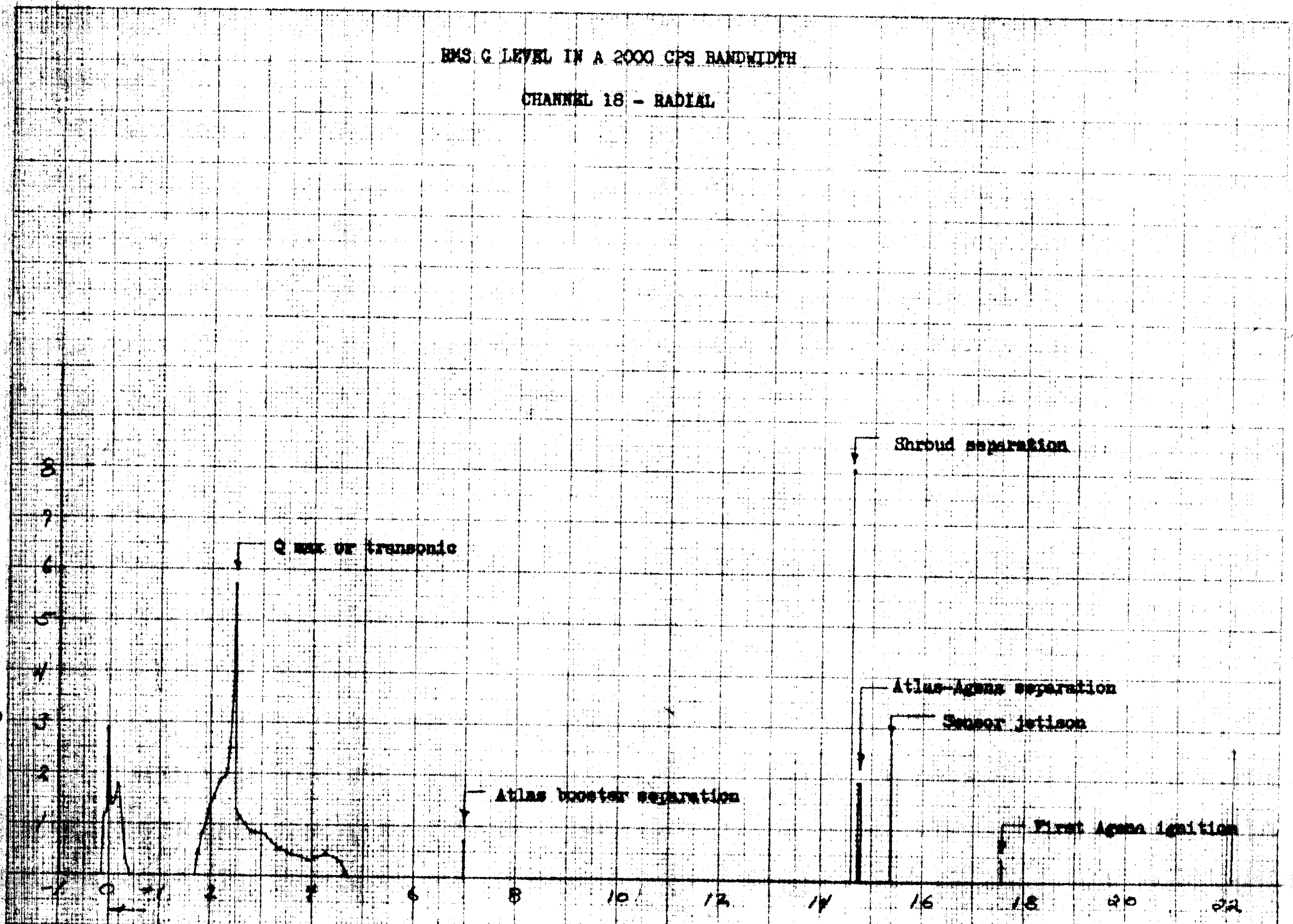
The composite acoustic-free response of the S/C to the measured input (Figures 3 and 4) is well down from the 30201 levels, indicating the test as adequate. The high levels on Figure 3 may have been increased due to acoustically induced vibration and could conceivably exceed assembly specifications. The probability of this happening is not significant since this would require a mechanical admittance match of the S/C to the acoustic admittance of the assembly. Considering all the possibilities, it is relatively safe to say that all assemblies and the S/C in general were adequately tested in the frequency range from about 1 to 1500 cps in all levels of test except the pitch and yaw directions of the PTM and FA of the S/C. This situation was anticipated some months ago and has been corrected for Ranger vehicles beginning with RA-3.

The shock environment, as represented by the sharp high level spikes in Figure 1 and 2 and the shock spectrum (Figure 7), have been adequately simulated by (1) the shock test as performed as part of the environmental test (top curves in Figure 7), and (2) actual firing of all of the flight type pyrotechnics on the Flight or Proof Test $M_o$del of the S/C.

The shock and vibration environment encountered by RA-1 S/C was within anticipated limits. The flight levels indicated for RA-1 were similar to flight levels encountered in Atlas-1 Agena's. There is one marked difference in the vibration encountered in the two configurations, this being the time at which the maximum vibration occurred. The previous flight of similar vehicles, the maximum wideband vibration level occurred at or near lift-off. In Ra-1, the maximum levels occurred during Q max. or transonic. The situation may be due to the aerodynamic shape of the Ranger vehicle



Relative Time of Flight - fig. 1



Relative Time of Flight - fig 2

RA-1 FLIGHT DATA

418C-

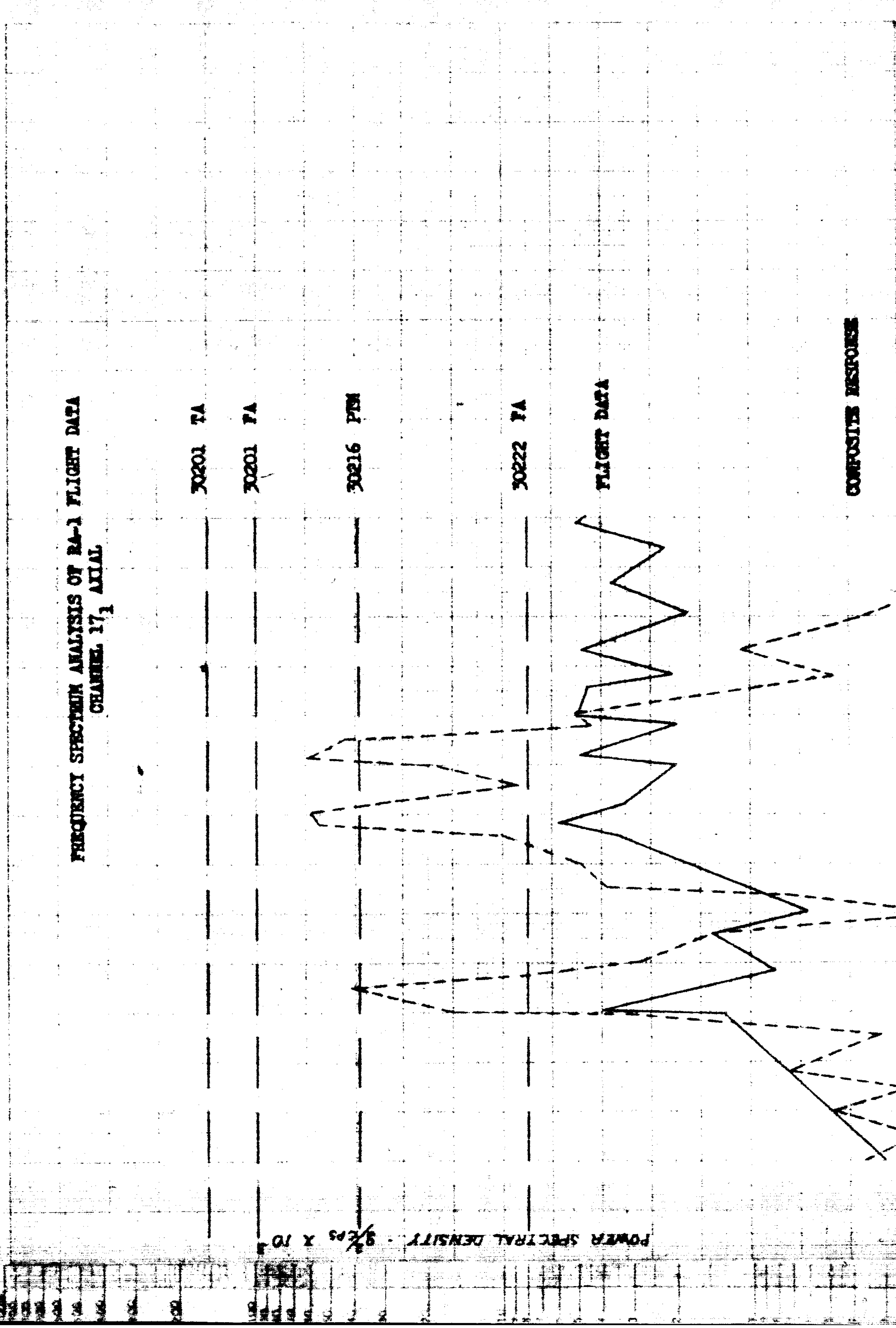


FIGURE 3

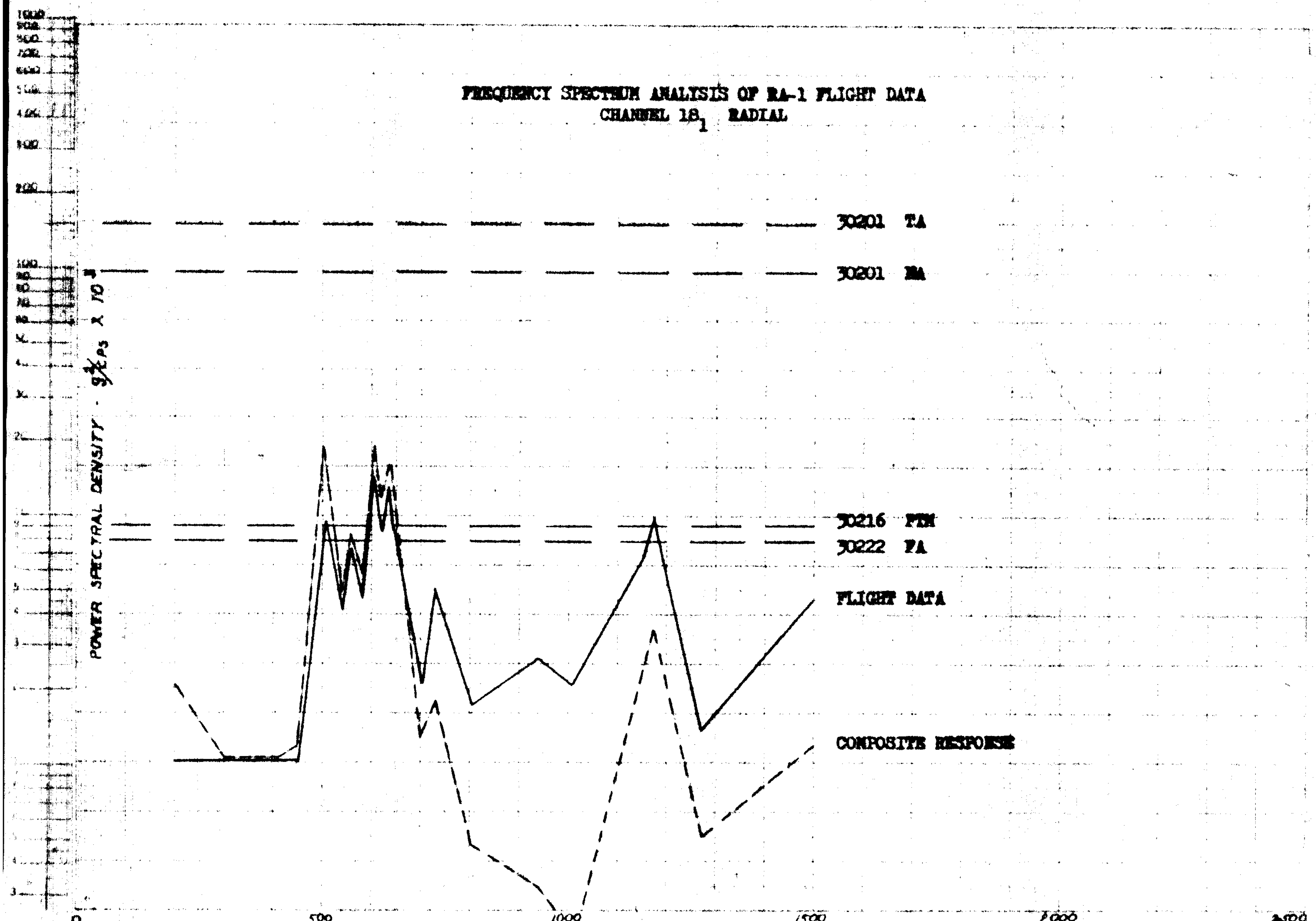

FREQUENCY SPECTRUM ANALYSIS OF RA-1 FLIGHT DATA
CHANNEL 18_1 RADIAL
30201 TA
30201 RA
30216 PTM
30222 FA
FLIGHT DATA
COMPOSITE RESPONSE
POWER SPECTRAL DENSITY - $g^2/cps \times 10^{-3}$
FREQUENCY, CPS
0
500
1000
1500
2000
2500

FIGURE 4

COMPARISON OF LOW FREQUENCY TEST LEVELS AS DESCRIBED IN JPL 30222

(SINE PORTION ONLY) WITH FLIGHT VIBRATION OF RA-1 S/C - PITCH AND YAW

RA-1 FLIGHT DATA

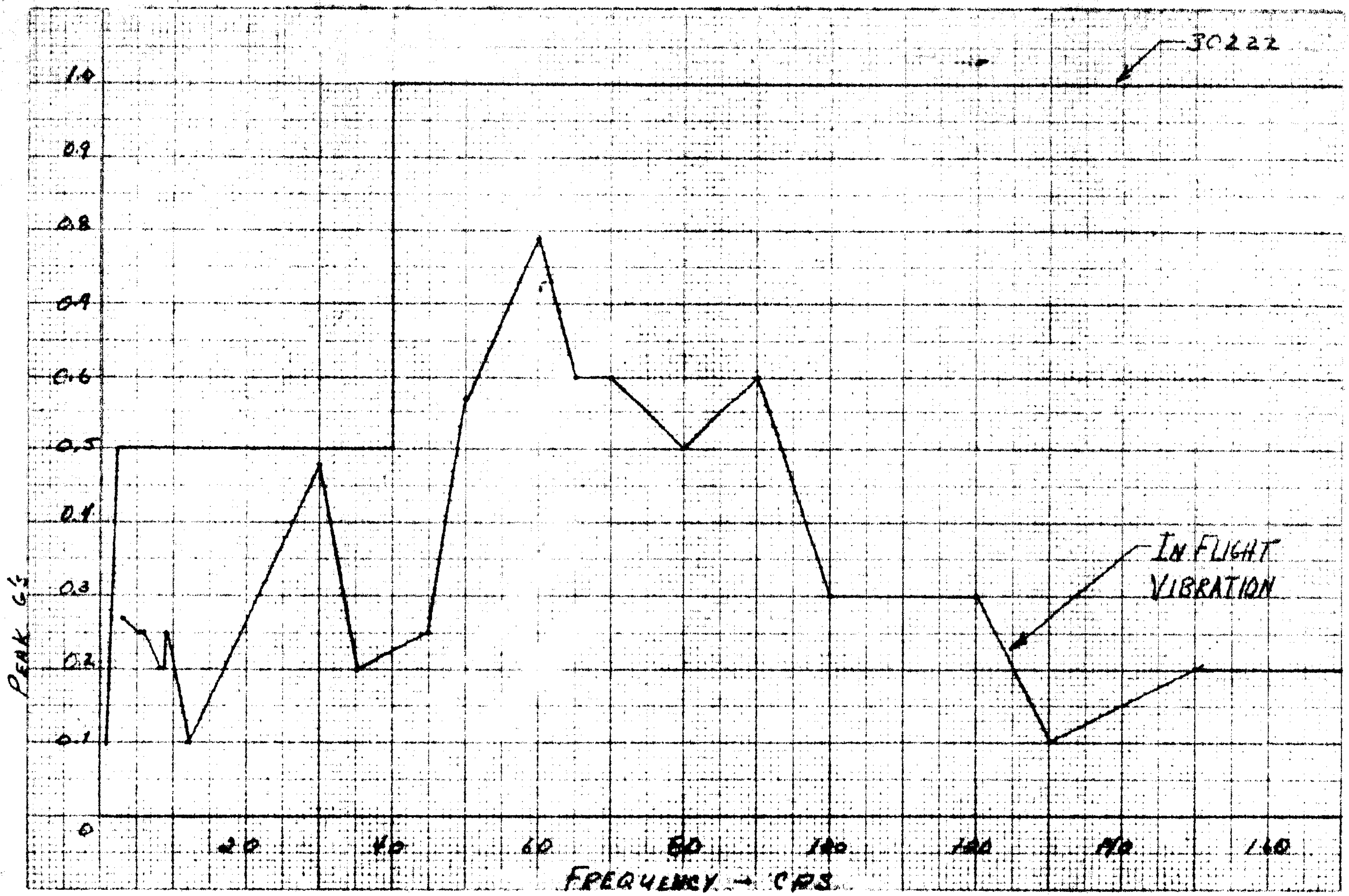


Fig. 5

COMPARISON OF TA VIBRATION TEST 30201 (SINE PORTION ONLY) WITH FLIGHT VIBRATION (Figure 5) MULTIPLIED BY THE MAXIMUM S/C GAIN CURVES

RA-1 FLIGHT DATA

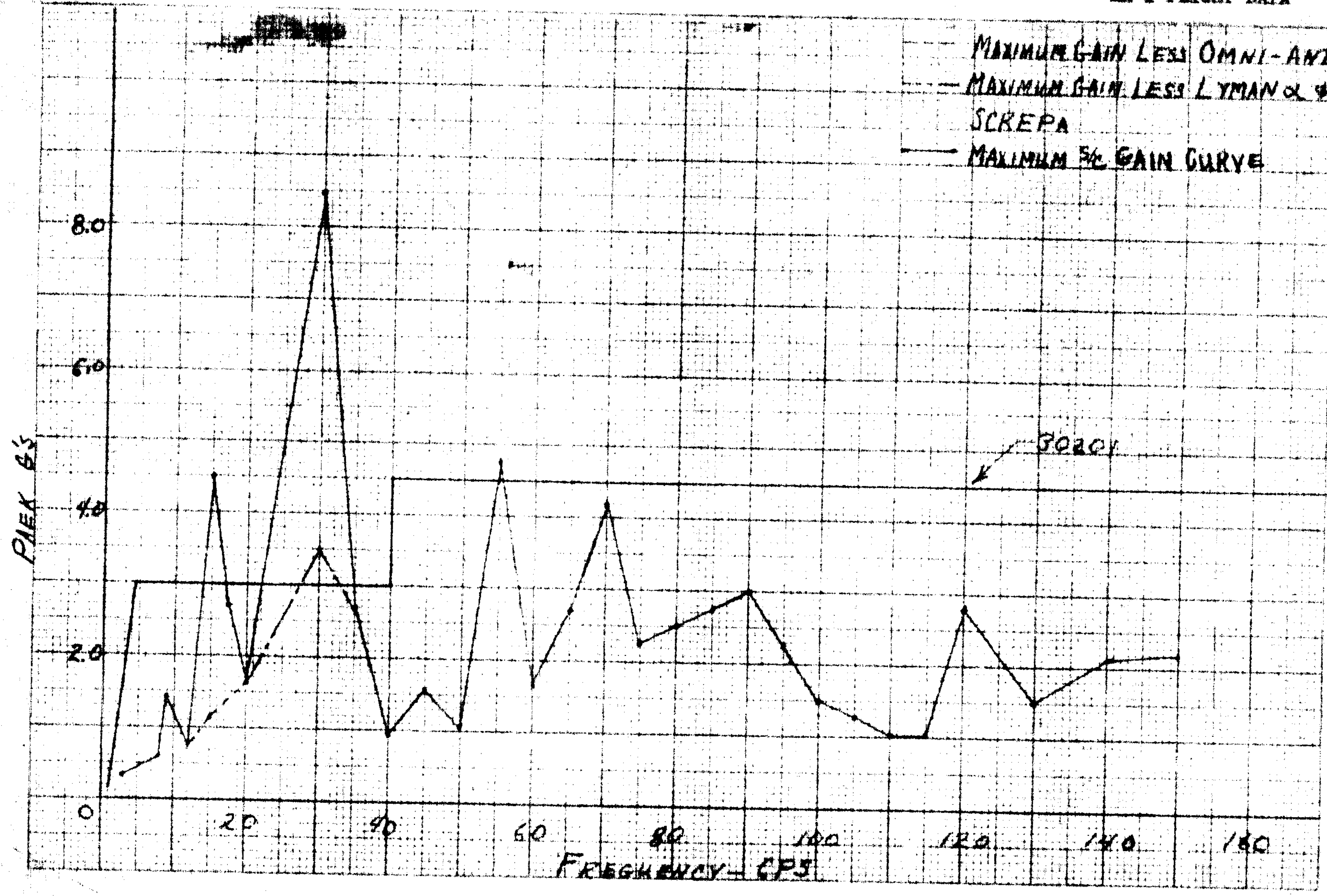


Fig. 6

THEORETICAL SHOCK SPECTRUM OF SHOCK TEST
FOR EQUIPMENT NOT CONTAINED WITHIN THE BUS

— MAXIMUM ENVELOPE OF SHOCK SPECTRUM FOR
TRANSIENT EVENTS FROM LIFT-OFF TO 1ST
AGENA CUT-OFF.

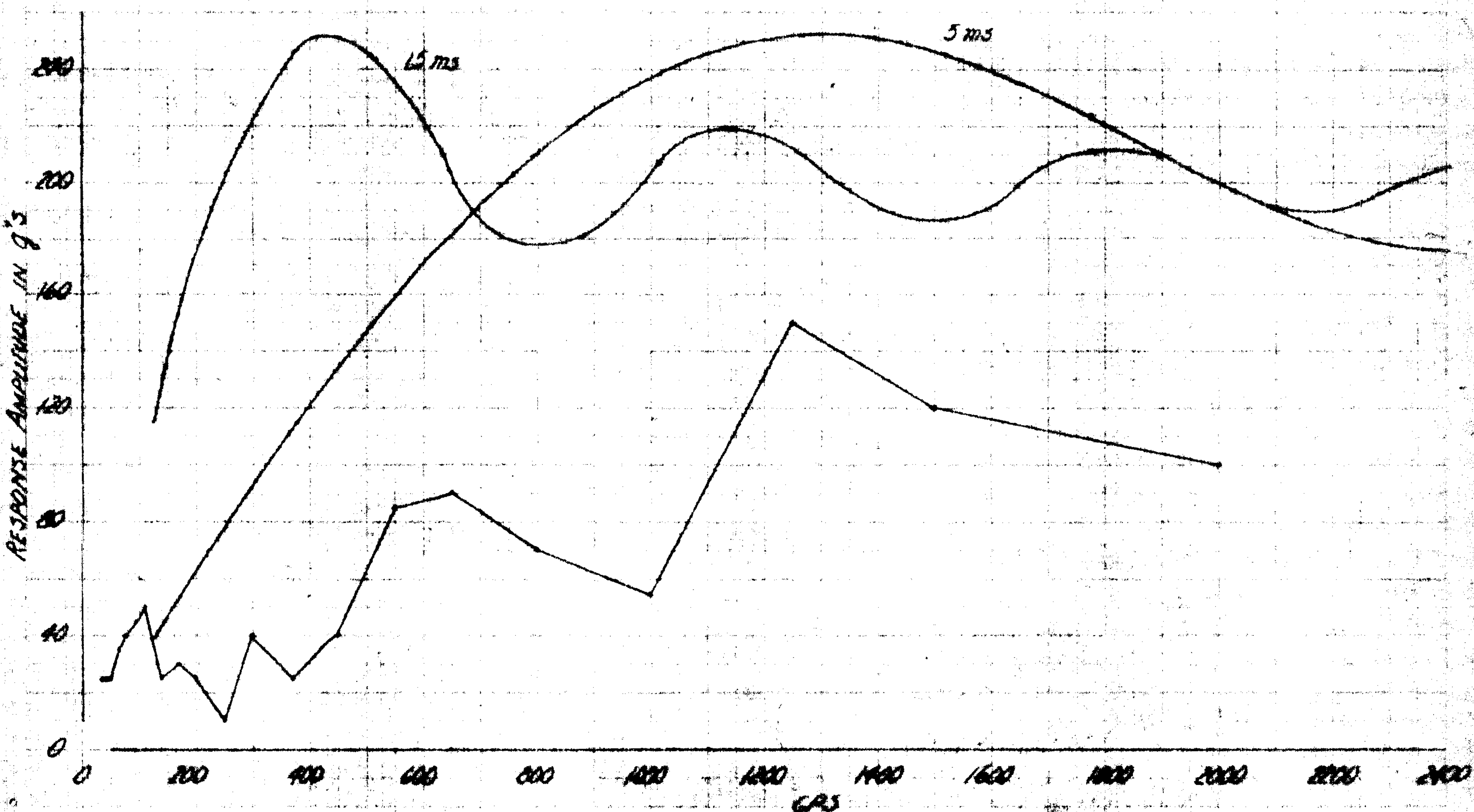
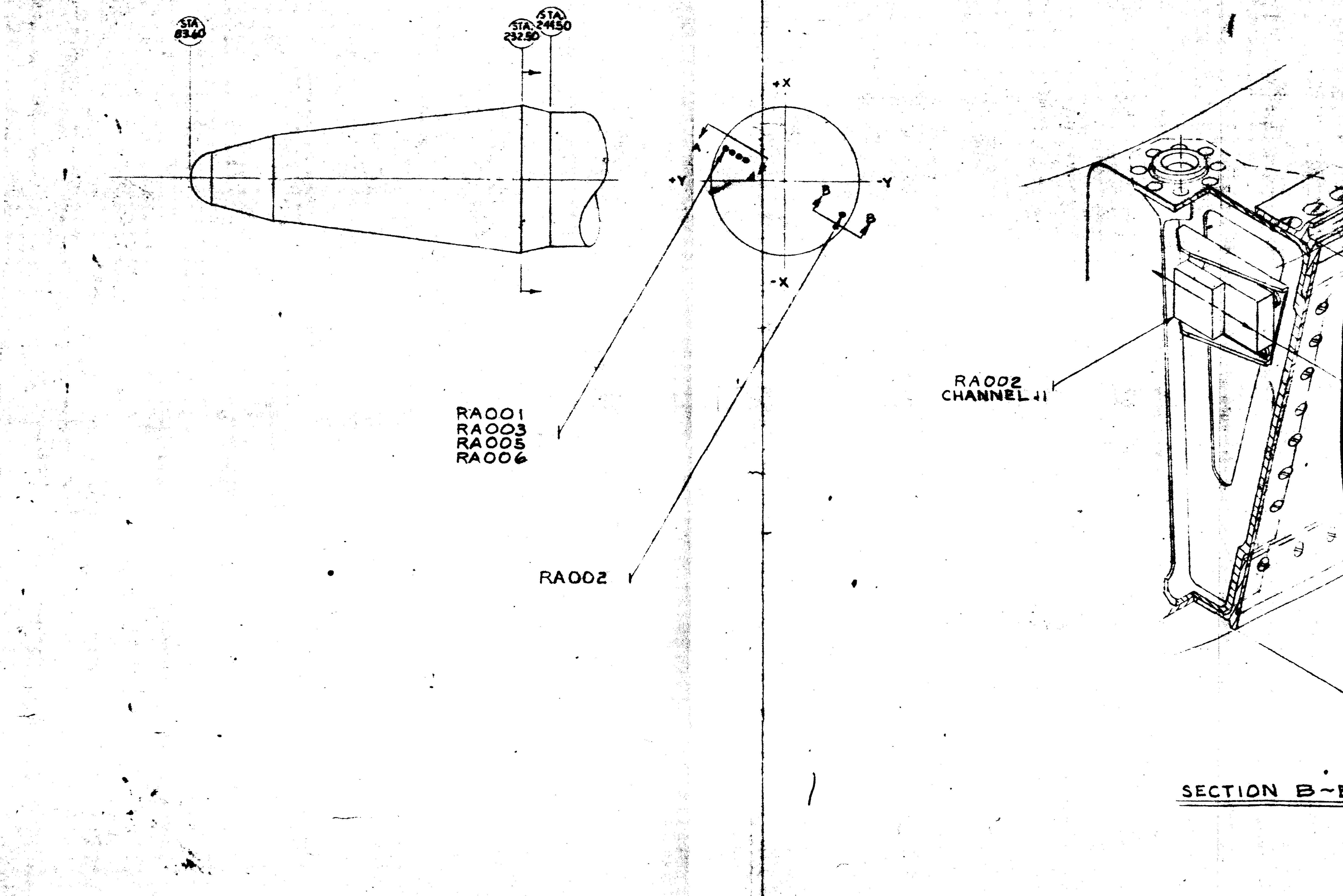
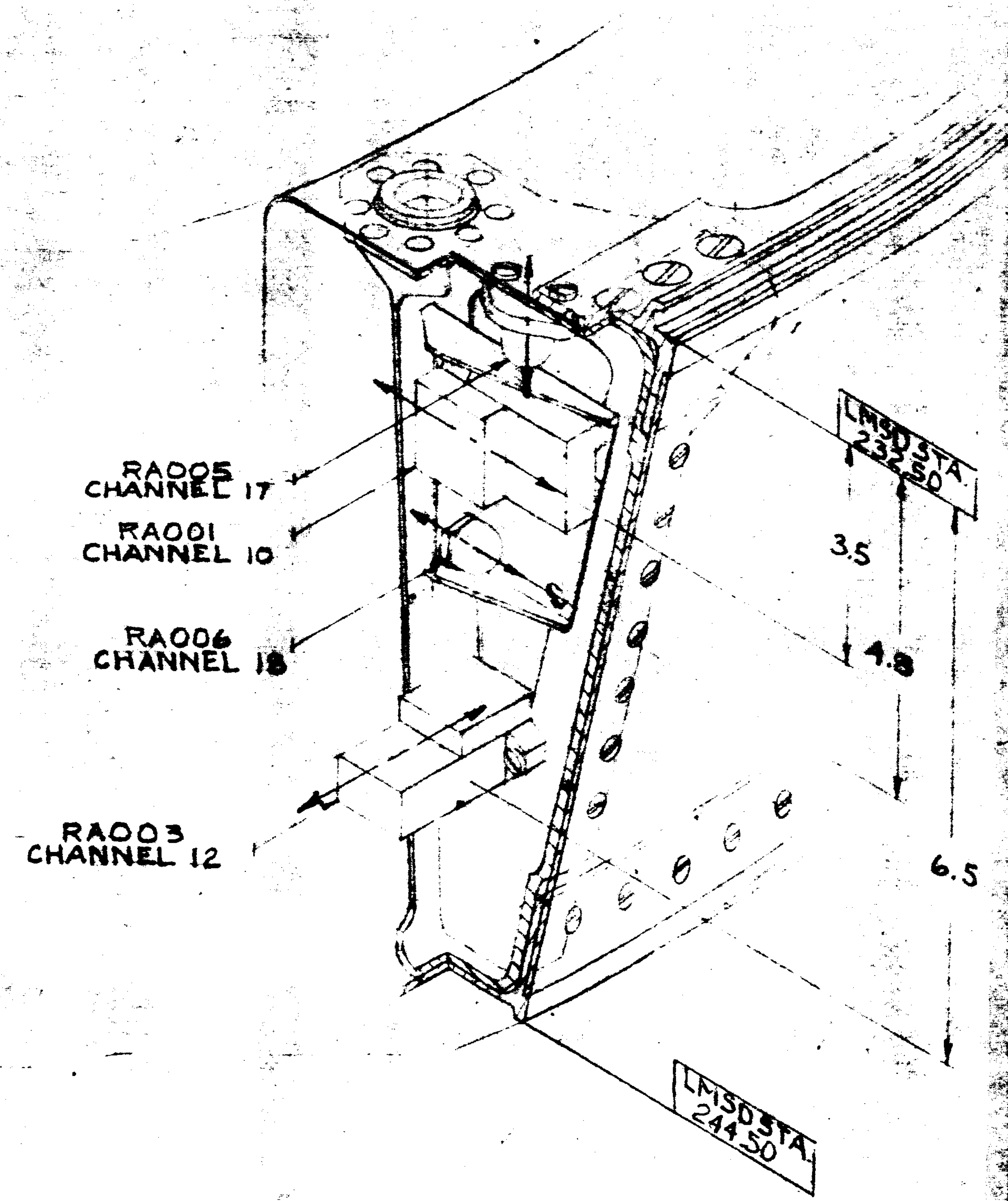


Fig. 7


STA 83.60
STA 232.50
STA 244.50
+X
-X
+Y
-Y
A
B
RA001
RA003
RA005
RA006
RA002
RA002
CHANNEL 11
SECTION B~B



SECTION A–A

INSTRUMENT INSTALLATION
AGENA ADAPTOR
R.G. MATHEWS 10-6-'61

2

<u>RA-1 SPACECRAFT SEPARATION</u> by J. I. McPherson - Division 35

Data transmitted from the three linear displacement potentiometers located on the LMSC adapter beneath Spacecraft feet B, D, and F indicated positive separation of the spacecraft from the Agena.

The nominal travel of each potentiometer was 2.5 inches. Actual travel of each potentiometer was measured at the time the spacecraft was mated to the Agena. Furthermore, a total travel distance was obtained from the telemetry record. The following is a table comparing the two sets of numbers.

	Actual Spacecraft Measurement of Total Travel inches	Total Travel From Telemetry Record inches
Channel 10	1.788	1.882
Channel 11	1.804	1.920
Channel 12	1.827	1.911

The actual spacecraft measurements were used in the reduction of the data.

Figure A shows a displacement versus time plot for the three potentiometers. Numerical differentiation of the above data shows that the velocity of the spacecraft relative to the Agena at the end of potentiometer travel was 1.5 fps.

The difference in the displacement readouts of the three potentiometers at any time was on the order of the accuracy of the instrumentation, so no exact pitch or yaw rates could be derived from the above data. It can be concluded, however, that the rates were less than the specified 3°/sec and that the separation was smooth.

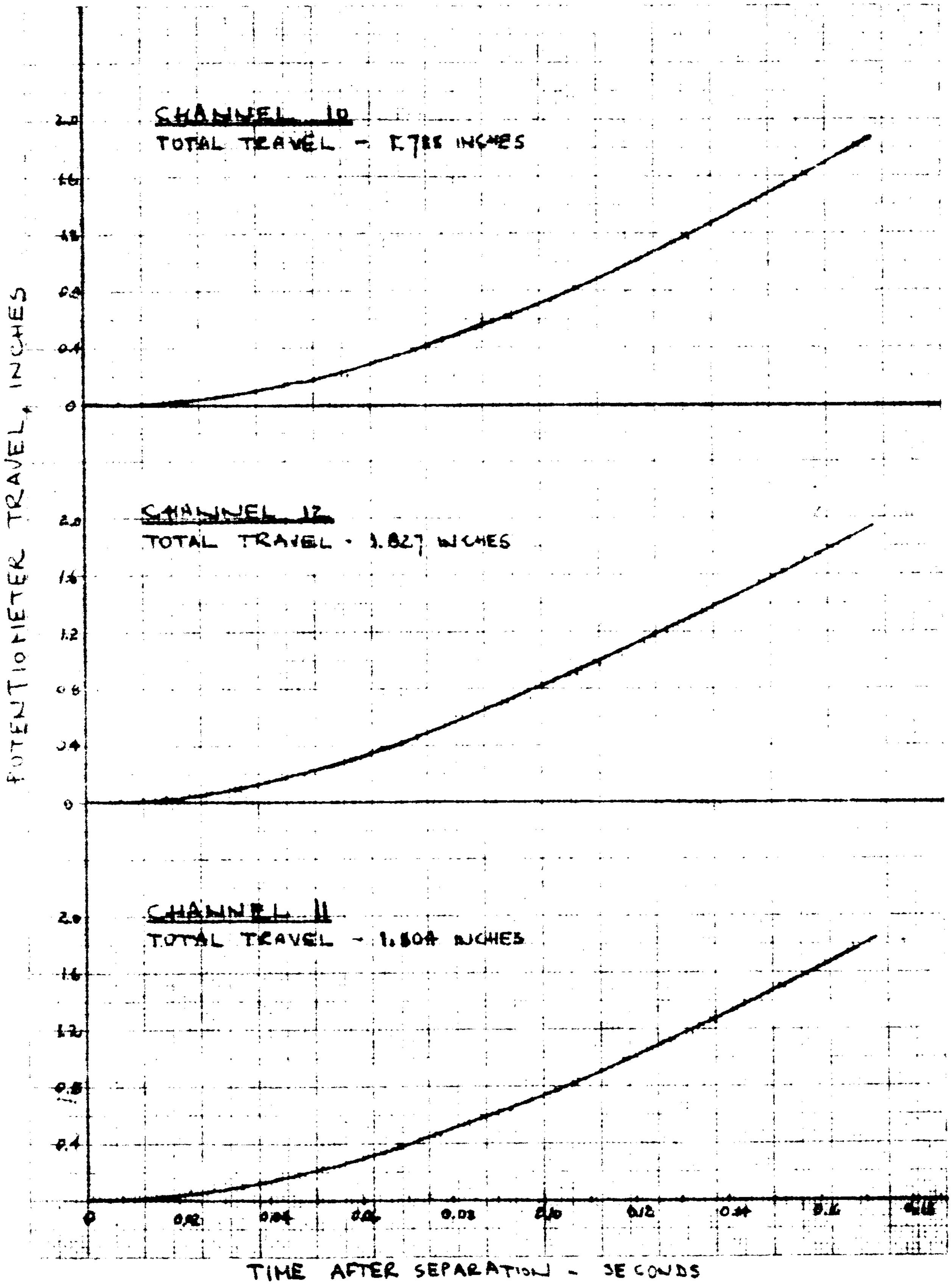

CHANNEL 10
TOTAL TRAVEL - 1.788 INCHES
CHANNEL 12
TOTAL TRAVEL - 1.827 INCHES
CHANNEL 11
TOTAL TRAVEL - 1.804 INCHES
POTENTIOMETER TRAVEL, INCHES
TIME AFTER SEPARATION - SECONDS

FIG. A

SOLAR PANEL AND BOOM ACTUATION by J. I. McPherson - Division 35

Solar Panel and Boom actuations were to be indicated by four blips. The first indicating that the controller had sent the "open" command and the second, third, and fourth signaling that the boom, minus x panel, and plus x panel respectively, were fully extended.

However, during the actuation period the MTS was in lock only intermittently and only three blips were received. These are believed to be the blips indicating that the boom and panels were fully extended. Assuming that the controller issued Command 3 at the proper time, actuating times and estimated bulk actuator temperatures were as follows:

	Actuating Time Seconds		Actuator Temperature
	Predicted	Actual	$^{\circ}F$
Boom	100	88	+10
-x Panel	150	127	+15
+x Panel	190	188	-5

That the solar panels opened is inferred from three additional pieces of information. First, the operation of the attitude control system indicates that the panels did not interfere with the operation of the sun sensors. Second, the solar panel temperatures were equal. Third, solar panel currents were maximum at the same time.

From the above it is concluded that the boom and panels were extended and that the operation of the actuation systems were as expected.

<u>RA-1 TEMPERATURES</u> by W. A. Hagemeyer - Division 35

Ranger I's highly unstandard flight path necessitated the recalculation of all temperature estimates based on the following assumptions:

A. <u>Stabilized</u>

1. Spacecraft had acquired the Sun, Earth Sensor had locked on the Earth, placing Cases II and III facing the Earth at all times.
2. Spacecraft in the shadow of the Earth 40% of each orbit.
3. One orbit lasts 90 minutes.

B. <u>Tumbling</u>

1. Spacecraft in the shadow of the Earth 40% of the time.
2. One orbit lasts 90 minutes.
3. Spacecraft tumbling at a constant rate about all axes.

In order to account for the effect of the Earth and sunlight reflected from the Earth, appropriate geometrical view factors and average values for albedo and Earth radiation were estimated. The geometrical configurations in themselves limit the accuracy of the results to an indication of trends only. Aerodynamic heating was included in the calculations, but it too is only a very rough estimate. Values of the above items at apogee and perigee were averaged.

The Hex, the Magnetometer, the Ion Chamber, the Solar Panels, and the Earth Sensor were analyzed for both the stabilized and tumbling conditions.

Average temperatures were calculated for the Hex, the Magnetometer, the Ion Chamber, and the Earth Sensor since these items have a high enough thermal inertia to remain fairly constant throughout an orbit. These are presented below in tabular form, along with flight temperatures.

	STABILIZED		TUMBLING	
	Calculated	Flight	Calculated	Flight
Hex	115°F	102°F - 115° F	123°F	118°F - 122°F
Magnetometer	142°F	140°F - 149°F	144°F	140°F - 158°F
Ion Chamber	50°F	50°F	39°F	50°F
Earth Sensor	150°F	130°F	130°F	110°F

The flight temperatures of the Magnetometer are based on the assumption that the transfer circuitry in the base of the Magnetometer is warmer than it would be in space; i.e., approximately +30°F to +70°F instead of -60°F to 0°.

Temperatures shown for the Hex in flight include the Triple Coincidence Telescope measurements.

Calculated temperatures for the Earth Sensor do not include the heat loss by conduction to the antenna. This loss is not easily estimated but is in the direction to lower the calculated temperature. However, the magnitude of the change between oriented and tumbling conditions agrees well with the measured flight temperatures.

Upon plotting the flight temperatures for the Hex, all readings take the jump from sun oriented temperature to tumbling temperature sometime between 1800 GMT on August 24, 1961, and 0200 GMT on August 25, 1961. Fairing in points sets the time at approximately 2200 GMT on August 24, 1961. The Triple Coincidence Telescope on top of Case VI follows a pattern similar to the Hex, with a temperature of about 113°F in the sun oriented phase and about 149°F during tumbling.

The Solar Panels respond much too rapidly to try and calculate any average temperature. Calculations show that during a sun oriented orbit, the panels will vary from -50°F to +140°F during the sunlit and shaded portions of the orbit respectively. Unfortunately, flight data does not correlate well with the upper temperature calculated. By determining the local time of each reading, it can be seen that very few of the readings are at a time in the orbit when the panels would be at either extreme of temperature. In this light, most of the points are at least in the right ballpark. Apparently we do not have good enough information about the surface properties of the panels to accurately predict their operation. At times when readings are received from both panels during the same pass, their temperatures are in agreement, indicating that the panels did open.

The best inference that can be made for the rest of the superstructure and experiments is that everything was running too hot, probably in the range of 86°F to 140°F. The two semi-conductor experiments and the Friction Package seem to bear this out.

In conclusion, it appears the Temperature Control system would have functioned properly in deep space and no changes are contemplated for Ranger 2.

<u>FRICTION EXPERIMENT</u> by J. B. Rittenhouse/L. D. Jaffe - Division 35

The friction experiment on RA1 performed as designed.

This statement is based on a thorough analysis of the analog data obtained from telemetry received by the Atlantic Missile Range launch telemetry trailer and the South Africa mobile tracking station from 0540 to 1539 GMT on 8/24/61.

The digital data obtained during the spacecraft pass over the MTS on 8/24/61 at approximately 1400 GMT established that the friction assembly was operating and that data on the coefficient of friction for specific materials could be correlated with the coding of the experiment commutator. The unreduced digital print-out from this pass was not itself of long enough duration to provide sufficient information to correlate the coefficient friction data completely, nor were the rest of the spacecraft passes over the other tracking stations. The code information built into the package was, however, fortuitously obtained on the 1400 GMT pass of 8/24/61 and this permitted the correlation of the data obtained from all other passes. On early records produced by the DRL, a decommutator sampling error resulted in digital information influenced by the slope of the rate-limited portion of the Rate 1 samples on the Channel 2 telemetry band.* Accordingly, analog information was used for the interpretation of the data.

If the spacecraft had experienced normal operation and if the ground decommutator remained in synchronization then unreduced digital information would be the desirable manner to present the friction experiment telemetry data to the cognizant engineer.

* Editor's Note: This situation was subsequently corrected.

The spacecraft in abnormal operation was not at altitudes where the vacuum of space was at the level desired to result in significantly lower vacuum than can be produced in the laboratory. The vacuum at 500 kilometers is of the order of 10^{-8} mm of mercury and at 100 kilometers the vacuum is to the order of 10^{-6} mm of mercury. These vacuums have been produced in the laboratory. Consequently, therefore, the coefficients of friction from the materials in the friction experiment of RA1 flight were not expected to be significantly different from those obtained with prototype equipment tested in the laboratory vacuum. This behavior was indeed true; RA1 flight data showed coefficients of friction for most of the materials slightly higher than in laboratory vacuum; the friction coefficients for the same materials were slightly higher in laboratory vacuum than in air.

The experiment was designed to be started by spacecraft controller command at 367 minutes after liftoff. Telemetry indicated that the experiment was not running at 322 minutes after liftoff. At the scheduled time (1611 GMT on 8/23) for controller command 9 to be given to start the experiment, the spacecraft was between passes over the mobile tracking station. On the following pass at about 1700 GMT the mobile tracking station was tracking the spacecraft beacon signal which carried no channel 2 telemetry. Therefore, it was not known from the telemetry if the friction experiment was started at or about the scheduled time. However, the experiment was running at about 0540 GMT on 8/24/61 during a pass over the Atlantic Missile Range tracked by the launch checkout trailer telemetry. There is no reason to doubt that start took place as scheduled.

The experiment was designed to operate for at least five hours with a 100% safety factor. On RA1 the experiment operated for at least 22 hours. The experiment was designed to turn itself off at a predetermined low voltage of the self-contained battery. It is believed that this event occurred sometime between the 1403 GMT and the 1539 GMT passes over the mobile tracking station on 8/24/61.

A more detailed report of coefficients of friction for material flown in RA1 and the comparisons of flight data with laboratory vacuum and atmospheric conditions data will be published in the SPS and in a JPL report.

The writer has been invited to present a paper with this data to the American Society of Lubrication Engineers in St. Louis, May 1962, with publication in one of the society journals shortly thereafter. If approval is obtained and the preprint deadline can be met, in all probability a report of the RA1 flight friction experiment result will be presented there.

GUIDANCE AND CONTROL by E. E. Suggs, J. Slay, P. C. Harrison, S. Szirmay - Division 34

Abstract

This report evaluates the performance of RA-1 on the basis of data telemetered from the spacecraft during its orbiting of the earth. Three subsystems are discussed, attitude control, power, and controller. Descriptions of the apparently normal operation of the spacecraft as well as of abnormal operation are presented.

1. Power Subsystem

A. Summary

Power system performance appears reasonable in view of the orbit achieved by the spacecraft; normal performance was impossible owing to the low earth orbit. The spacecraft experienced alternate light and shadow periods which precluded the normal function of the solar panels, (continuous conversion of sunlight to electrical energy.) The attitude control system was able to achieve solar acquisition in the light periods only during the first day of flight. As a result of these circumstances, the launch and backup battery supplied most of the power for the spacecraft. The 0.25 watt beacon battery appeared to have a lifetime less than expected, probably due to excessive heating of the spacecraft just prior to reentry. The power conversion and distribution system appear to have functioned as designed. The attitude control converter appears to have been operating in an overload mode on the second day of the mission. The overload was apparently removed on the third day at which time the converter resumed delivering power to the attitude control subsystem.

B. Solar Panels

Simultaneous current measurements from both solar panels occurred only once during the mission. Attitude control data observed at this time indicates that the spacecraft was solar oriented. The pertinent data is presented in Table 1.

TABLE 1

Date	Time	Measurement	Engineering Units
8/23	131205	Solar Panel 4A9 Current	3.3 amperes
8/23	131345	Solar Panel 4A10 Current	3.3 amperes
8/23	131355	Solar Panel 4A9 Temperature	$-18^{\circ}C$
8/23	135025	Primary System Voltage	27.4 volts
8/23	135205	Primary System Current	3.8 amperes

It will be assumed that the primary system voltage and current readings, although not received at the same time as the solar panel current measurements, indicate a value of system power typical of the values at this time. It is further assumed that both panels are at the same temperature, although the temperature is probably not stabilized at this time, and that the power demanded by the spacecraft is 104 watts (27.4 x 3.8). Field data on RA-1 has indicated that with a system voltage of 27.5 volts, the solar panel operating voltage is 29.5 volts. At this voltage, the current into both panel shunt diode networks totals approximately 3.0 amperes. The 2.8 ampere difference between total panel current of 6.6 amperes and system current of 3.8 amperes can thus be attributed to shunt diode current.

Each solar panel delivered 3.3 amperes at 29.5 volts for a total power of 195 watts. The nominally expected maximum power performance of two Ranger panels is 175 watts at $39^{\circ}C$, and approximately 215 watts at $-18^{\circ}C$. Assuming panels were at $-18^{\circ}C$ during the time that currents were measured, the expected performance was demonstrated within accuracies of the measurements.

Thus, it can be concluded from the observed data that the solar panels functioned as might be expected under the circumstances, the design power output having been reached. Satisfactory performance is thus indicated for a normal trajectory.

C. AD Solar Cell

The advanced development solar cell experiment was designed to measure short circuit current and open circuit voltage. The only current reading was obtained on 8/23 at 134905. A current of 18 ma was recorded compared to the expected value of 160 ma, indicating a partially shaded condition. Open circuit voltages of .245 and .255 volts were recorded on 8/23 and 8/24 indicating experiment temperatures of 40°C and 50°C respectively. The open circuit voltage reading is used to apply a correction factor to the short circuit current reading and thus was not useful for this flight. However, the temperatures recorded were within the expected case V temperature range.

D. Launch and Backup Battery

In evaluating the launch and backup battery it is assumed that the drain on the battery is relieved to a negligible degree by the solar panels. The capacity of the launch and backup battery was estimated before flight to be 8530 watt-hours. The observed power drain on the battery over the total flight period amounts to 10,770 watt-hours. Table 3 shows system voltage, current, and power by days.

TABLE 3

Date	System Voltage	System Current	Power	Remarks
8/23	28.6 v	3.9 a	111.5 w	At launch
8/23	27.3 v	4.0 a	109.2 w	Last pass of day
8/24	25.1 v	3.57 a	89.6 w	A/C Converter in overload all day
8/25	25.3 v	4.95 a	125.1 w	
8/26	25.2 v	4.8 a	119.0 w	
8/27	25.2 v	--	--	Assumed 4.8 a and 119.0 w
8/27	21.2 v	3.8 a	80.56w	Beginning of battery failure

The accuracy of the assumptions probably limits the estimate of battery capacity to plus or minus 1000 watt-hours. Hence, the actual capacity appears to have been in excess of the design estimates by at least 1200 watt-hours, and perhaps by as much as 3200 watt-hours. The battery temperature followed the bus temperature as was expected. Temperatures ranging from 80°F on 8/23 to 120°F on 8/27 were observed. This temperature range should have produced no severe ill effects on the battery performance.

E. Beacon Battery

The 0.25 watt transmitter battery (beacon battery) provided power to the beacon through the sixth day of flight, 8/28. The main battery failed on the fifth day. The temperature of the beacon battery is assumed to have followed the temperature of the bus, although no data is available after the main battery failed. If the temperature did follow the bus temperature, it is reasonable to assume that the beacon battery reached a temperature at which catastrophic failure would occur, i.e., (140°F to 180°F) sometime on the sixth or seventh day of flight, 8/28 or 8/29. Such temperatures would occur just prior to reentry, which is assumed to have occurred on the eighth day, 8/30. Hence, it is concluded that the observed life of the beacon battery is consistent with the orbit achieved.

F. Power Switching and Logic

The power switching and logic functions of the power subsystem appear to have operated as designed throughout the flight. The converter monitors indicated that power was being delivered to all users as intended during the second day of flight (8/24). On this day, the attitude control converter monitor voltage is zero, indicating that no power is being delivered to the attitude control subsystem. The attitude control measurements on this day indicate that power is being applied momentarily and removed. This is evidenced by a series of spikes at constant frequency each time the attitude control measurements appear on the telemetry records. This is the normal indication of an overload condition in the attitude control subsystem. Similar telemetry records have been generated by testing the PTM with a low resistance across the output of the attitude control converter. These tests are discussed in the Attitude Control section of the report. On the third day of flight (8/25) the converter returned to normal operation, and the abnormalities in the attitude control measurements disappeared.

The overload condition is a symptom of a malfunction in the attitude control subsystem or a manifestation of a power mismatch. There is a strong argument for the latter case. The PTM test also revealed that the attitude control system may demand as much as 1.2 amperes from the converter in normal operation. The overload point of the converter was set at exactly 1.2 amperes. It is quite possible that the attitude control system functioned normally and demanded an overload current from the converter.

Further discussion of the apparent failure is deferred to the Attitude Control section of the report.

2. Spacecraft Controller

A. Summary

Controller commands appear to have been given at the proper times within the limits imposed on the determination of command times by the available data. Timing functions of the controller were on schedule all during the flight except for the second day when malfunctions appeared in other subsystems.

B. Controller Commands

All controller commands, with the exception of command No. 3 (solar panel squib actuation) were issued when the DSIF was not tracking the spacecraft. Command No. 3 nominally programmed to occur 2200 seconds after timer start, was observed during the first pass over the Mobile Tracking Station in South Africa. There appeared to be a small discrepancy between the nominal time and the observed time of execution of this command. This discrepancy was not of a magnitude to indicate incorrect operation of the timer, due to a problem with the MTS time generator during this time period.(1)

Commands which could be "bracketed" were as follows:

COMMAND	NOMINAL TIME AFTER LAUNCH	BRACKETS
No.6 Rate Gyro Scale Factor Change	115 min 20 sec	91 min -- 128 min
No. 7 Antenna Switchover	247 min	232 min -- 318 min
No.8 Reduce Data Rate	363 min 40 sec	322 min -- 415 min
No.9 Friction Experiment	367 min	322 min -- 415 min(2)

C. Controller Timing Functions

The 1 pps. to both telemetry and science has shown no indication of malfunction. The 1 pulse per 1000 seconds records indicate some variation in the timing, both to telemetry and science, on the second day of flight. No such variations were noted at any other time. Since this is the time during which abnormalities were noticed in other subsystems, it is the present conclusion that noise introduced into the system by these malfunctions was causing extraneous pulses to be counted along with the 400 cps "clock" and resulting, ultimately, in the observed timing errors.

(1)The telemetry recorded during this pass was very noisy due to the low signal level. The event telemeter channel was below discriminator threshold at the programmed time of Command 3. See pg.20 for an interpretation of event blips recovered.

(2)This bracket is uncertain because no friction experiment telemetry was recovered during pass at L+415 min. Command No. 2 for science power up (nominal L+58.7 min) came between L+34 and L+87 min, according to information from M. Neugebauer ...
A. E. Dickinson

3. Attitude Control Subsystem

A. Summary

Solar acquisition appears to have occured on the first day for at least two passes through the sunlight. Earth acquisition is somewhat ambiguous due to the abnormally large earth. However, during the first day, while the spacecraft was in the sunlight, the earth sensor was excited by a light source, apparently the earth. The observed acquisition of the sun appeared to be consistent with the predicted operation of the spacecraft and consistent with the orbit achieved. The gas consumption was much higher than normal, but is explainable in terms of the abnormal orbit. The largest part of the effort in analyzing the data has been directed toward explaining the apparent malfunction mentioned in the Power Subsystem portion of this report. A detailed report on progress to date in this area is contained in Part E of the following discussion.

B. Solar Acquisition

Data from the tracking stations in South Africa and from Goldstone on the first day of the mission indicated for at least part of the time that the roll axis of the spacecraft was pointed toward the sun, the pitch and yaw axes having been successfully attitude-stabilized. During the period when this data was observed the solar panel currents were at the design level, further supporting the conclusion regarding successful sun acquisition. The time differential of the telemetered position measurements at this time indicated turning rates of about six times the normally expected rates after acquisition. Brief analyses show that torques as great as one-forth the gas jet control torques may act on the spacecraft due to atmospheric effects in the low orbit. Torques of this magnitude would easily account for the increased rates; even higher rates would not be unusual in such an environment.

Data from at least one pass indicates that normal acquisition did occur as the spacecraft entered the sunlight but was later lost, perhaps due to these abnormally large torques. It is to be concluded from the initial analysis of the sun-acquisition data that the attitude control system functioned as designed within the limits imposed by the trajectory.

C. Earth Acquisition

Data taken at the same time as that which described sun acquisition indicates that the earth sensor was tracking a lighted object. Actual operation is in doubt owing to the large size of the earth. It is presumed that the measurement of roll position is of little value because the angle subtended by the lighted earth at the spacecraft is more than twice as large as the total field of view of the earth sensor. The rate gyro observations at the time indicate that the roll rates were controlled within the designed capabilities of the attitude control system.

D. Gas Consumption

It is estimated from the limited number of gas-bottle pressure measurements obtained that the gas supply was exhausted within about ten hours after injection. Based on gas consumption calculations performed prior to the flight, it is estimated that one sun acquisition and one earth acquisition would use about 0.3 pounds of gas. In ten hours six or seven sun and earth acquisitions would be initiated as a result of the alternate sunlit and shadowed periods. Seven acquisitions, the number to be expected in a 10 hour period, would use 2.1 lb of gas.

It is estimated that one pound of gas would have been used in 40 days of normal cruise operation: however, spacecraft angular rates observed when the vehicle is presumed to have been attitude stabilized would increase the normal cruise gas consumption by a factor of about 30 to 40, so that 0.5 lb of gas would be used in about twelve hours. These estimates show how the total supply of 2.5 lb of gas would be used in a 10-hr period, without considering the gas used in overcoming initial rates, atmospheric torques, and transients during the transition from light to dark. Hence, it appears that the gas supply lasted somewhat longer than would be anticipated in the abnormal environment.

E. Discussion of Abnormalities

1. Introduction

An abnormality which affected both the power system and the attitude control system occurred on the second day of flight. As explained in Part F of the section of the Power Subsystem, the attitude control system was not receiving power on the second day and the attitude control converter was in an overload mode of operation. Both prior to and following the second day, no indication of this condition was observed. No evidence of component failure has been found. Both before and after the occurrence of the abnormality, the attitude control system appeared to be operating as expected under the ambient conditions.

The number of malfunctions that can cause power loss for relatively long periods of time and then disappear with no apparent permanent failure, is quite limited. Of the possible causes for the observed system operation that were postulated, most can be placed into one of two categories: a momentary overload caused the converter to go into an overload mode from which it could not recover owing to the high initial power transient of a full system start, or a partial line short existed throughout the second day which was cleared by the high current interrogation pulses of the converter. The existence of a partial rather than a complete short circuit is indicated by the appearance of the interrogation pulses on the attitude control telemetry records.

It seems very likely that the attitude control system was presenting, in either case, a large power demand because the system had exhausted its gas supply, was probably far from any attitude of solar acquisition, and probably tumbling rapidly. Such a condition would require a relatively high current from the power supply. The overload setting of the converter prior to launch was 1.2 amperes, which was 165% of predicted maximum load. Some further postulates and some test results, described below, make use of this likely condition of large power demand.

2. Tests and Test Results

Test A. Several tests were performed on PTM modules to evaluate the first possibility. First, measurements of sub-system line currents were made to confirm previous measurements. Plus and minus line currents were measured for full tumble and idle conditions and were found to be appreciably greater than previous tests. The results were:

E	I
+28 v	.78 amp*
-26 v	.58 amp**
+28 v	.62 amp***
-26 v	.42 amp***

* Conditions

1. All ccw valves actuated
2. Antenna driving off positive supply
3. Earth sensor operating
4. Gyros delivering full torque from plus supply

**

1. All cw valves actuated
2. Antenna driving off negative supply
3. Earth sensor operating
4. Gyros delivering full torque from negative supply

***

1. No valves actuated
2. Sensor signals at null (position & rate)

Proper interpretation of these measurements yields a total line current as sensed by the overload circuit of 1.20 amps - - the value at which it was set to trigger prior to flight.

Test B. In the next test the attitude control system was excited to place full power demand on the converter. The overload circuit was set at 1.2 amperes and a test was performed to determine the ability of the converter to recover from an overload under full load conditions. In conducting this test an external load was added to cause the converter to fall into overload. The load was then removed and the recovery observed. In each case, independent of the degree of overload, the converter recovered on the removal of the externally applied load. On the basis of results of these two tests it was concluded that the overload was of a sustained rather than temporary nature and that the converter recovered immediately on removal of the overload. The first test, however, indicates that the marginal overload setting of the converter is a likely cause of power loss.

Test C. The second series of tests were made to provide a better understanding of the converter characteristics in overload. In Test C, the converter was loaded by the Attitude Control System and additional resistive load was provided to place the converter in the overloaded mode. Observations of the peak interrogation pulse current, wave form and recurrance rate were noted and are described below.

1. The peak pulse current varies directly with the applied load up to a maximum of 1.6 amps for a short circuit on one line. For short circuits on both lines, the peak current is equally divided, and the current on each line is .8 amps.

2. The interrogation pulse wave form varied as a function of overload, advancing rapidly from that shown in Figure a. for very light overloads to that of Figure b. for moderate and heavy overloads.

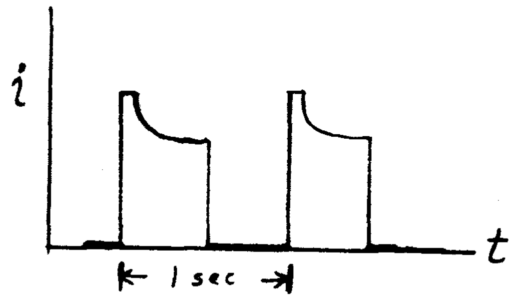


Figure a. Light Overload

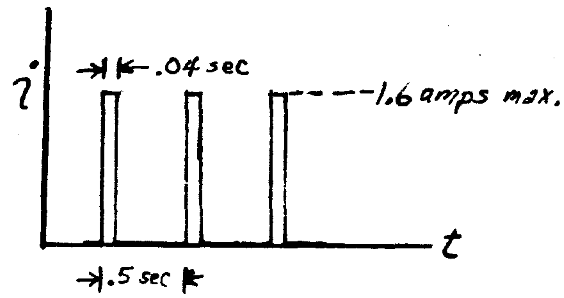
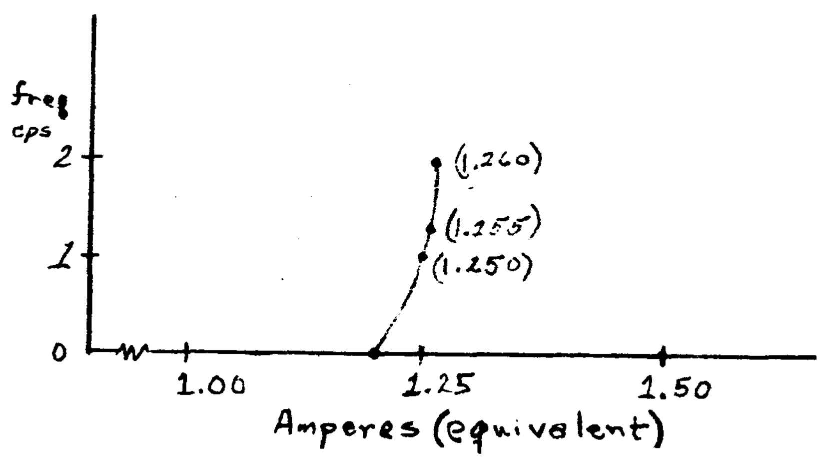


Figure b. Moderate to Heavy Overloads

3. As indicated above the interrogation frequency also changed as a function of load from approximately 1 cps to 2 cps. A plot of this is shown below.



Test D. A test to determine the length of time an overload must exist before the overload circuit would operate was desired. Since this test is difficult to implement, indications of the results of such a test were sought from previously obtained data. Such an indication is available from Test C-2, Figure b., wherein under short circuit conditions the interrogation current remained for 40 milliseconds before the supply was disabled. It is therefore concluded that overload durations of less than 40 ms will not cause a power loss and those in excess will disable power.

Test E. Since the interrogation pulses were observed on flight telemetry data of the rate gyro outputs, similar observations were made in the laboratory to determine if an indication of line current could be obtained from pulse amplitudes as observed at the rate gyro outputs. For this test the gyros were torqued at maximum rate and the output wave form and amplitude was noted. Curves of these are shown below.

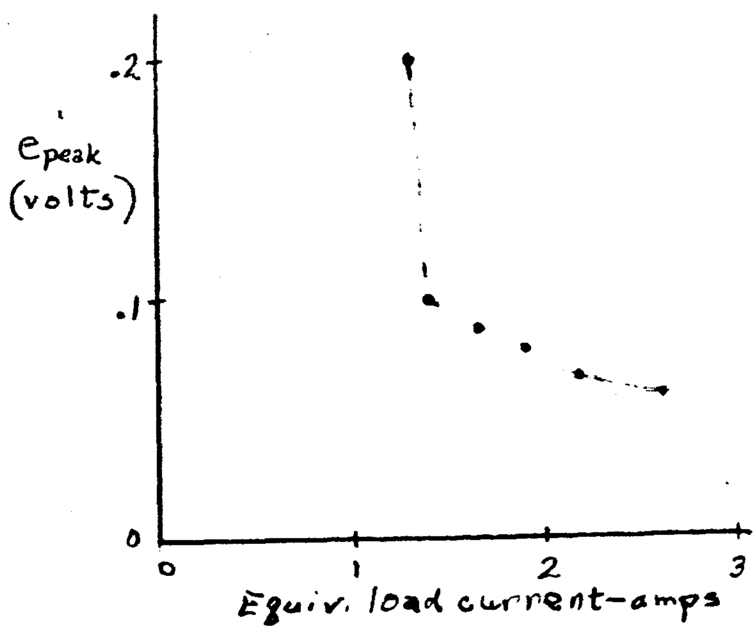


Figure a.

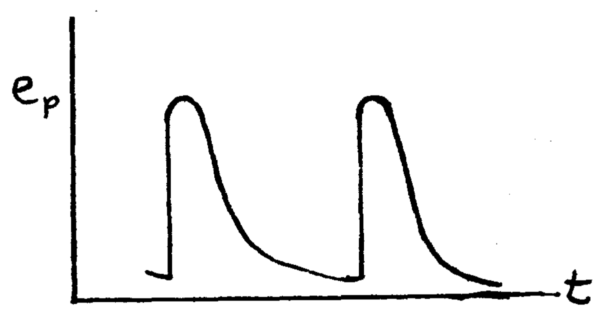


Figure b.

This data has not yet been compared with flight results

Test F. The third sequence of tests examined the Attitude Control System for possible short circuit paths which could have been cleared on the third day by the converter interrogation pulses. Each module was examined and the following observations were noted.

1. Switching Amplifier:

   Aside from capacitor failures the output stages of the switching amplifier could have overloaded the converter if a short circuit occurred between actuator valve terminals. The interrogation pulses could have cleared the short by opening the output stage junction. Past experience, however, indicates that this was an unlikely occurrence in that accidental valve short circuits have invariably resulted in an open circuit failure of the affected output transistor.

2. Gyro Electronics:

   It is unlikely that a failure could have occurred in this module in that the outputs of each gyro were telemetered. Had a short circuit occurred in any of the output circuits (the only likely locations), the interrogation pulses would not have appeared in the telemetry records.

3. Antenna Drive module:

   No circuit conditions were found which would yield the observed performance.

4. Earth Sensor:

   Several areas were investigated to determine the effects on earth sensor performance and power supply loading for the abnormal trajectory. These are described in an IOM from T. Baxter to V. Anthony dated October 11, 1961. In general, no serious loading effects were observed due to high lighting or possible corona conditions. A failure in the output stage could have caused excessive converter loading, accounting for the second day power loss. Since proper Earth Sensor operation was obtained on the third day this is not a likely cause of the second day power loss.

III. Conclusions:

A thorough check of the Attitude Control System under conditions expected during the second day has not revealed any evidence of a component failure. At the present time, the most logical explanation for the power loss is simply that the converter overload trigger point was not set high enough to handle the absolute maximum power demand of the system. Recovery on the third day may be explained by increasing temperatures observed in telemetry records which relieved the converter load by increases in valve and gyro torque resistances.

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